

**A HYDROGEN-OXYGEN ROCKET ENGINE COOLANT PASSAGE DESIGN
PROGRAM (RECOP) FOR FLUID-COOLED THRUST CHAMBERS AND NOZZLES**

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SUMMARY

The design of coolant passages in regeneratively cooled thrust chambers is critical to the operation and safety of a rocket engine system. Designing a coolant passage is a complex thermal and hydraulic problem requiring an accurate understanding of the heat transfer between the combustion gas and the coolant. Every major rocket engine company has invested in the development of thrust chamber computer design and analysis tools; two examples are Rocketdyne's REGEN⁽¹⁾ code and Aerojet's ELES⁽²⁾ program. In an effort to augment current design capabilities for government and industry, the NASA Lewis Research Center is developing a computer model to design coolant passages for advanced regeneratively cooled thrust chambers. The RECOP code incorporates state-of-the-art correlations, numerical techniques and design methods, certainly minimum requirements for generating optimum designs of future space chemical engines. A preliminary version of the RECOP model was recently completed and code validation work is in progress. This paper introduces major features of RECOP and compares the analysis to design points for the first test case engine; the Pratt & Whitney RL10A-3-3A^(3,4) thrust chamber (Fig 1).

MODEL DESCRIPTION

The RECOP model generates a preliminary design of tubular and milled channel coolant passages in regeneratively cooled thrust chambers given basic engine performance and geometry design constraints: thrust; chamber pressure; mixture ratio; expansion and contraction area ratio; characteristic chamber length; nozzle percent length; initial divergence and convergence half-angle; coolant flow path; coolant passage type; coolant inlet conditions; axial location of coolant inlet; hot-gas side wall temperatures; and chamber wall materials of construction. Output from the code provides information relevant to conducting detailed design and optimization studies of the engine: thrust chamber and nozzle dimensions and contour; wall heat flux; coolant pressure drop; coolant heat pick-up; coolant passage dimensions; combustion gas and coolant thermal and transport properties; and engine weight. A thrust chamber design schematic with typical dimensional requirements is shown in figure 2. The numerical procedure for the RECOP model is summarized in figure 3 and the main engine parameters computed by RECOP are listed below:

- Specific impulse and characteristic exhaust velocity (ODE)⁽⁵⁾
- Thrust chamber and coolant mass flow rate
- Thrust chamber and nozzle size and contour
- Combustion gas properties (ODE)
- Hot-gas side heat transfer coefficient and heat flux
- Coolant-side wall temperature and coolant total enthalpy
- Coolant transport and thermodynamic properties (GASP)⁽⁶⁾
- Coolant Reynolds and Mach number
- Coolant viscous and momentum pressure drop
- Coolant-side heat transfer coefficient and heat flux
- Coolant velocity by iteration until heat fluxes balance
- Final thrust chamber coolant passage dimensions
- Thrust chamber weight

Fundamental rocket engine performance and design equations^(7,8) are used to compute thrust chamber size, throat diameter and chamber contour. The Rao method⁽⁹⁾ generates the nozzle contour; a parabolic approximation for a Bell nozzle. There are options in RECOP for designing conical nozzles and entering nozzle coordinates specified in the form of a polynomial. The main calculation routine divides the engine into axial segments in the direction of coolant flow. The one-dimensional heat flux is calculated given the design hot-gas side wall temperature distribution. The Bartz equation provides the hot-gas side heat transfer coefficients. Starting at the coolant inlet a Newton-Raphson iterative technique on the coolant-side provides heat transfer coefficients, coolant velocity and coolant passage dimensions. A solution is obtained at each axial station when the coolant-side heat flux is within a prescribed convergence tolerance.

There are five coolant flowpath configuration options available in the RECOP program: counter-flow; parallel-flow; split-flow; one-and-one-half pass; and series-flow. Insulating and gradated coating layers on the thrust chamber wall may be specified. RECOP is limited to engine designs ranging from 20 to 5000 psia chamber pressure, 2 to 12 mixture ratio and 2:1 to 1500:1 nozzle expansion area ratio. The preliminary version of RECOP was restricted to hydrogen/oxygen (H/O) and liquid hydrogen coolant primarily because of the availability of design information for H/O systems. Additional propellant combinations and coolant fluids will be incorporated into subsequent versions of RECOP. The program was written in FORTRAN-77 on a DEC-VAX computer; a PC version of the RECOP code is in the development phase.

RL10A-3-3A ANALYTICAL RESULTS

Figure 4 shows the RL10A-3-3A design parameters used in the RECOP code. Analytical thrust chamber contour results (Fig 5) agree with design coordinates from the injector face to an area ratio of about 8:1; the predictions are below design along the axial coordinates of the mid-nozzle region due to the Rao approximation. The hot-gas side design wall temperatures (Fig 6), hot-gas side

heat flux (Fig 7) and coolant static temperature profiles (Fig 8) all show good agreement with design information. The coolant static pressure results (Fig 8) deviate from the engine design data especially in the throat region. One reason for the large difference is that the "design" coolant static pressures shown in figure 9 were from a Pratt & Whitney analysis, but more importantly they contradict actual engine test data. Coolant pressure drop data from engine tests ranged from 260 to 280 psia; the RECOP code predicted a coolant pressure drop of 256 psia. Regarding coolant passage full-length tube geometry, a comparison of calculated tube height and width versus engine design points indicates good agreement (Fig 10). An overall comparison of RECOP with eighteen RL10A-3-3A design points showed that 39 percent of the predictions deviated from design by less than 1 percent; 33 percent were within 2 percent; and 28 percent were greater than 2 percent.

REFERENCES

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FIG 1 : RL10A-3-3A ROCKET ENGINE



FIG 2 : REGENERATIVELY-COOLED THRUST CHAMBER AND NOZZLE CONTOUR

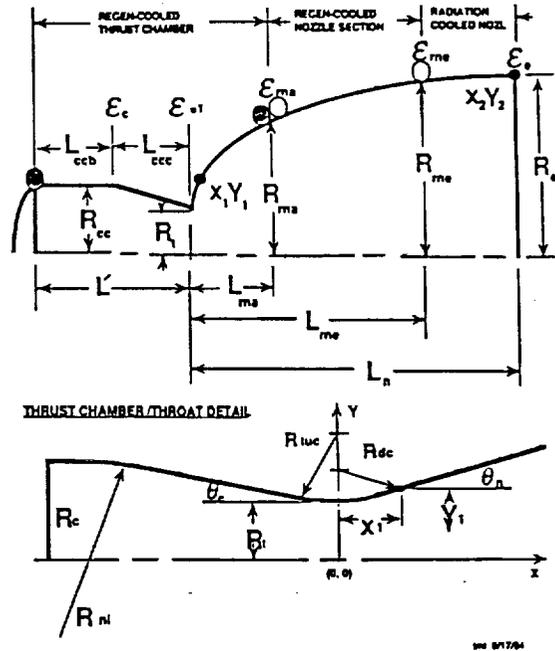


FIG 3 : COOLANT PASSAGE DESIGN CODE FLOW CHART

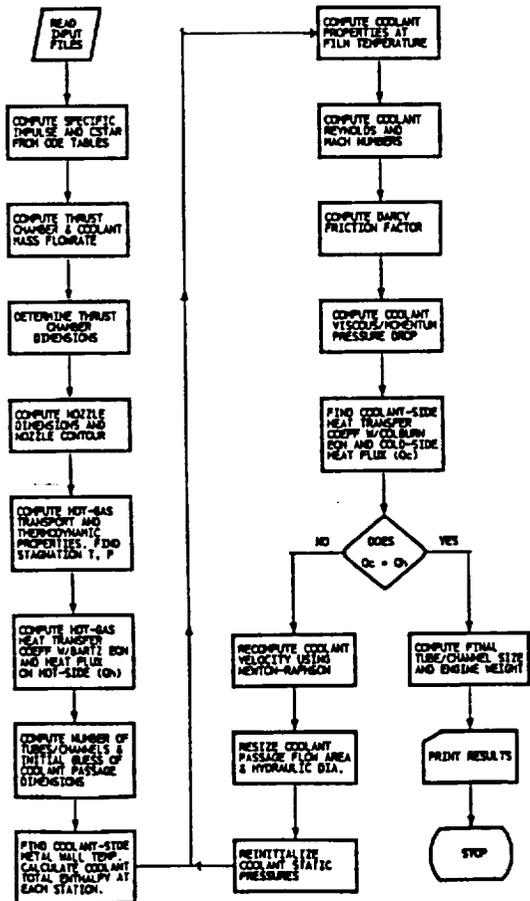


FIG 4 : RL10A-3-3A THRUST CHAMBER DESIGN INPUT PARAMETERS

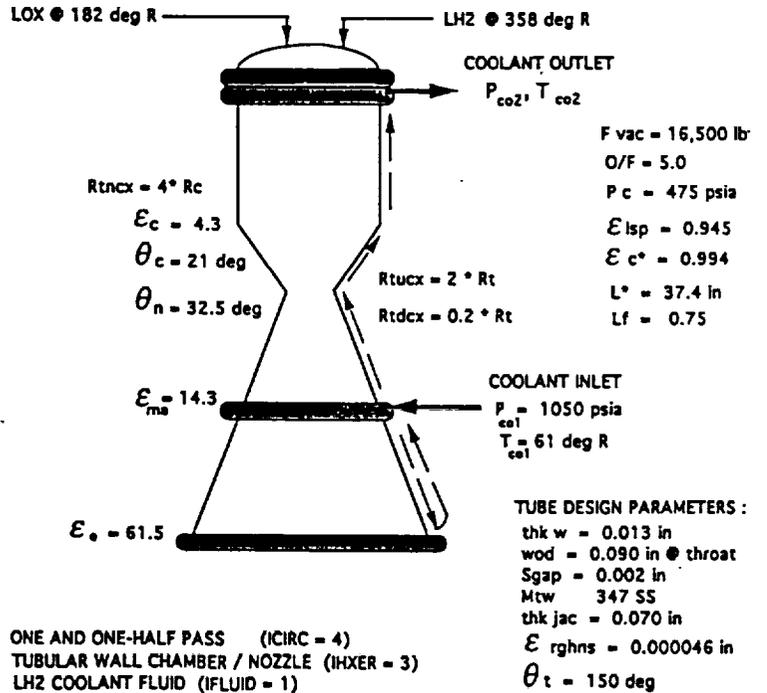


FIG 5 : RL10A-3-3A CHAMBER WALL RADIUS PROFILE

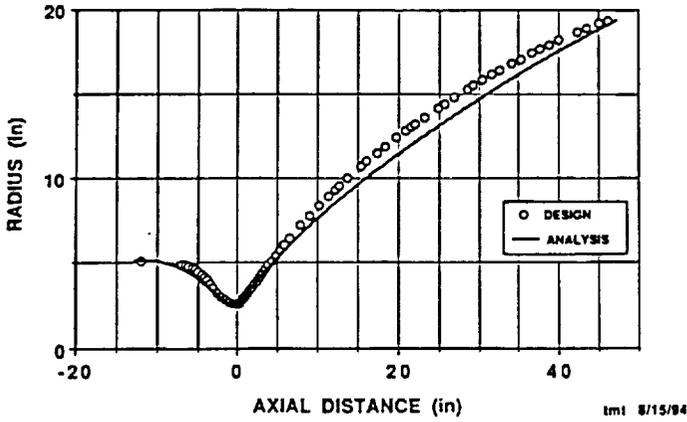


FIG 6 : RL10A-3-3A HOT-SIDE WALL TEMPERATURE PROFILE

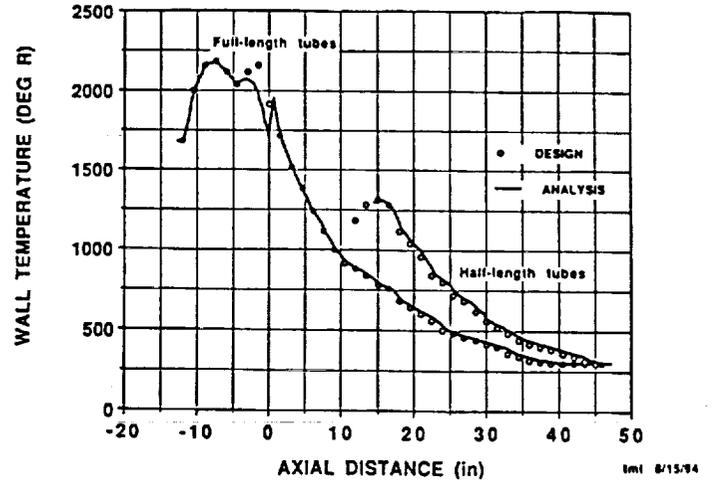


FIG 7 : RL10A-3-3A WALL HEAT FLUX PROFILE

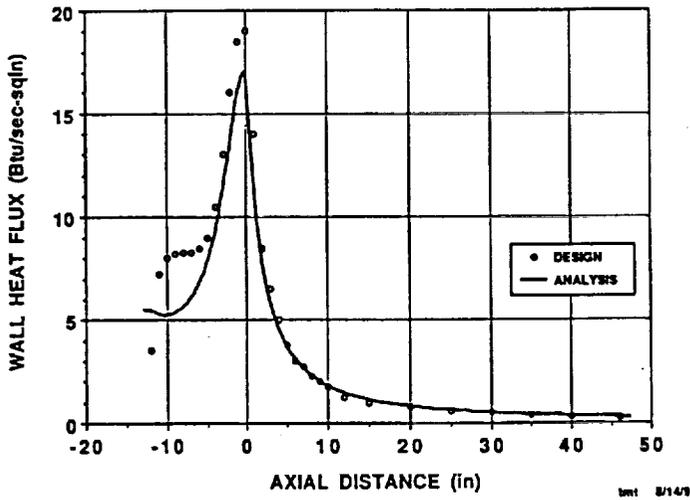


FIG 8 : RL10A-3-3A COOLANT STATIC TEMPERATURE PROFILE

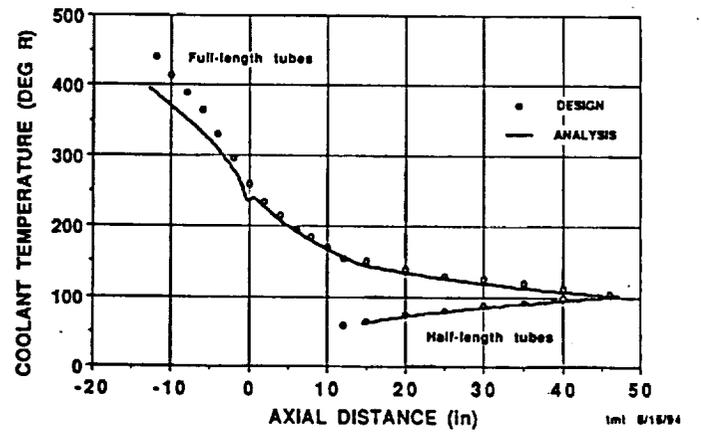


FIG 9 : RL10A-3-3A COOLANT STATIC PRESSURE PROFILE

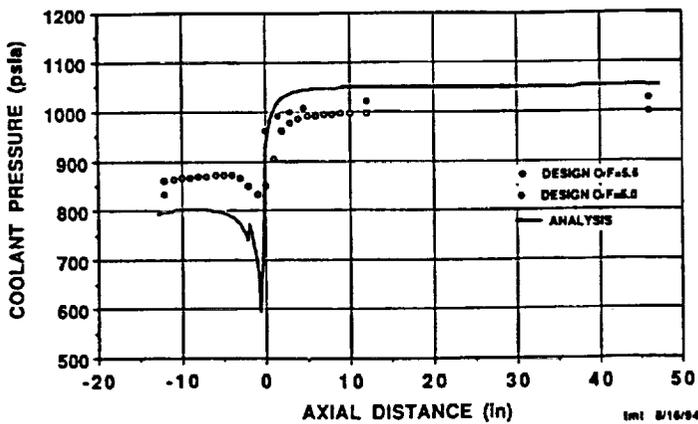


FIG 10 : RL10A-3-3A TUBE WIDTH AND HEIGHT PROFILE

